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Space Shuttle Main Engine

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ABSTRACT

The Space Shuttle Main Engine (SSME) is a reusable, high performance rocket engine being developed to meet the performance, life reliability, and operational requirements of the Space Shuttle. Significant engine features include a staged combustion power cycle developing chamber pressure in excess of 3,000 psia, high area ratio nozzle expansion, throttling capability, and a computer operated engine control system. The SSME is currently undergoing certification testing at the National Space Technology Laboratoreis focusing on demonstrating maturity and reliability for manned flight this year. Current status regarding engine performance, system characteristics, and test results will be summarized. A comparison of the SSME development and certification programs with engines successfully used in the Saturn Program will be presented.

INTRODUCTION

Many years of planned technology advancement in large liquid propulsion engine components is paying off for numerous engineers in government and industry, who during the last decade have worked on the many challenges facing the designer of a reusable high performance LOX/hydrogen rocket engine. Today's Space Shuttle Main Engine design, rapidly approaching flight qualification status, encompasses to a large degree the output of the last decades planning in liquid propulsion technology and materials research. This engine, combined with other propulsion elements of the National Aeronautics and Space Administration's Shuttle vehicle, will provide ascent thrust at high specific impulse for the nation's Space Transportation System. Three (3) engines will be used in a clustered configuration with ignition occurring on the launch pad and extending for approximately nine (9) minutes into powered flight. The more significant operating characteristics and features of the engine are depicted in Figure 1.

Each engine provides 2.09 X 10^6 N (470,000 LBS.) of thrust at rated conditions during altitude operation. High performance at these thrust levels is obtained by operating the engine's main combustion chamber at 2065 N/cm^2 (2995 psia). This operating combustion pressure, combined with a nozzle expansion ratio of 77.5:1, is designed to provide a nominal vacuum specific impulse of 4464 N-s/kg (455.2 seconds). The engine will be capable of operating up to 109% of rated thrust conditions during select Shuttle flights depending on mission payload requirements, with throttling capability to 65% of the rated thrust level. Reusability and extended life is a significant feature of this engine compared to prior large rocket engine programs. The design life goal to be demonstrated during the certification programs will be directed toward establishing 7.5 hours of operational life and 55 missions prior to any need for extensive refurbishment.

Design and development of the Space Shuttle Main Engine is being conducted by the Rocketdyne Division of Rockwell International under contract to the George C. Marshall Space Flight Center. This development activity was initiated in 1972 and has now progressed to the point where extensive certification testing for the first Shuttle launch is underway at the National Space Technology Laboratories (NSTL) located in Mississippi. Figure 2 depicts one of three (3) single engine test facilities used in the development program. During the course of the program sixteen (16) new engines have been built and tested, including the three (3) flight engines scheduled for use on Orbiter Vehicle 102 (OV102) currently located at the Kennedy Space

Center. These flight engines were delivered in July of 1979 and are shown in Figure 3 installed in the vehicle. Since the initial single engine test in May of 1975, approximately 530 tests have been conducted accumulating in excess of 65,000 seconds of operating time. Several successful cluster engine tests have been conducted adding another 5,000 seconds of engine test exposure at vehicle environmental conditions. Single engine development testing, as well as clustered engine testing, during the past several years has focused on exposing the engine components, control system, and inter-related subsystems to the limiting conditions expected on the initial Shuttle flights. In the course of this development activity, various technical problems, typical of prior engine development programs, have been encountered but appear at this time to have been satisfactorily resolved. The more significant problems that were encountered during development testing include (1) establishing proper turbopump bearing load sharing, (2) developing dynamic seal durability in the rotating turbomachinery, (3) defining turbine blade fatigue limits, (4) eliminating any fretting of piece parts in oxygen systems, and (5) establishing the dynamic load environment in critical engine ducts and main injector oxidizer post elements. To establish through test demonstrations that these problems have been solved, an extensive certification test program has been established and certification testing in support of the first Shuttle launch is well underway. Two certification engines, of the flight drawing configuration, have now successfully demonstrated 12,500 seconds of the required 20,000 seconds of operation at conditions enveloping the specification requirements over the range of 65% to 100% of rated thrust levels. This testing is currently scheduled to be completed late this summer. In parallel to the single engine certification tests, the three-engine cluster Main Propulsion Test Article (MPTA) will be completing the remaining objectives that integrate all major liquid propulsion elements of the Shuttle Vehicle. Included in this test article are the three (3) Space Shuttle Main Engines, the Orbiter propellant feed system, the Shuttle Vehicle External Tank, and other Orbiter subsystems supporting these elements.

MAJOR DESIGN FEATURES

The Space Shuttle Main Engine utilizes the staged combustion cycle with 100% of both propellants, oxidizer and hydrogen, being burned in the main combustion chamber prior to expansion through the 77.5:1 area ratio nozzle. The physical assembly of the major components in the engine are depicted in Figure 4.

The four turbopumps, two low pressure and two high pressure, are the key elements in establishing the system operating parameters. Both low pressure pumps are connected to the vehicle propellant ducting and supported in a fixed position by the vehicle structure. The discharge of each low pressure pump is connected to the inlet of the high pressure pumps by flexible ducts providing engine gimballing capability and thrust vector control throughout powered flight. Both low pressure pumps are axial flow machines and operate at relatively low speeds providing the necessary net positive suction pressure to the inlet to the high pressure turbomachinery. The high pressure fuel turbopump is a three-stage centrifugal pump driven by a two-stage hot gas manifold. The discharge of this turbopump supplies liquid hydrogen to the preburner assemblies and the nozzle and combustion chamber coolant circuits. The high pressure oxidizer turbopump consists of two (2) centrifugal pumps (main pump and preburner pump) on a common shaft driven by a two-stage hot gas turbine. The main pump supplies oxygen to the main chamber injector and to the preburner pump where liquid oxygen is routed to the preburner assemblies.

The hot gas manifold is the structural backbone of the engine supporting the two preburners, high pressure pumps, main injector, and main combustion chamber. The hot gas manifold further interconnects the fuel and oxidizer preburners to the main combustion chamber. Both preburners are welded to the hot gas manifold and generate the low mixture ratio, fuel rich, combustion products that power the high pressure turbomachinery. Both preburners consist of a combustor with a single pass, fuel cooled jacket and a baffled coaxial-element injector. The injector of the main combustion chamber also employs baffles and the coaxial-element design. The injector face plate is cooled by gaseous hydrogen. The gimbal bearing of the engine is bolted to the main injector dome assembly and is the thrust interface between the engine and vehicle allowing the entire assembly to gimbal for vehicle thrust vector control.

The main combustion chamber is a cylindrical, regeneratively cooled, structural chamber that contains the burning propellant gases and initiates the expansion from the chamber throat to an expansion ratio of 5:1. The chamber is flange attached to the hot gas manifold and consists of a regeneratively fuel-cooled Narloy-Z (copper alloy) coolant liner and a high strength nickel base alloy structural jacket. The nozzle is bolted to the main combustion chamber and is constructed of tapered tubes reinforced with jacketed sections and hatbands. The nozzle assembly is a regeneratively fuel-cooled, 80.6% bell chamber that completes the expansion of the main combustion chamber gases from a 5:1 to 77.5:1 expansion ratio.

The engine controller is attached to the main combustion chamber by clevis fittings. The controller provides redundant computers and associated internal electronic assemblies for total system control and monitoring of all engine functions during checkout and engine operation. The controller assembly, in conjunction with engine sensors, valves, and actuators provides (1) closed loop thrust and propellant mixture ratio control, (2) engine start and shutdown sequencing, (3) engine flight readiness verification, (4) engine performance limit monitoring, and (5) response to vehicle commands and transmission of engine status, performance, and maintenance data to the vehicle.

DEVELOPMENT AND CERTIFICATION TEST STATUS

With the successful completion of acceptance testing of the three (3) flight engines, conduct of extensive test exposure on one certification engine, and further testing on a second certification engine currently in progress, sufficient data is now available to assess reliability of the engine configuration planned for the first Shuttle launch. Six (6) complete engines of this configuration have been built and tested, all during the past 12 months, and incorporate the most recent engineering changes. Engine serial numbers 2005, 2006 and 2007 assigned to flight use, engine serial number 0008 assigned to cluster engine vehicle testing, and engine serial numbers 2004 and 0009 currently undergoing extensive certification testing comprise this flight configuration data base. These engines have been collectively tested 71 times and accumulated 17,711 seconds of operation under simulated flight conditions. This test exposure, considering a three-engine vehicle cluster, equates to 23.7 mission start-stop cycles and 11.35 missions of nominal vehicle firing time. From this data base, engine reliability and performance under flight conditions can be assessed; a summary of which is provided in Figure 5. Highlights of this summary are as follows: (1) reliability in completing the cluster engine countdown without extensive holds or rollback, .916, (2) reliability of a successful cluster engine start sequence with total control system redundancy on all engines up to the point of SRB ignition, .873; a comparable reliability value if redundancy of instrumentation involved in flight redline parameters is not required would be .958, and (3) the reliability of completing the ascent burn with all engines firing the total scheduled duration is .912.

Of the 71 tests included in this data base, 40 of these tests have been conducted on certification engine serial number 2004. No significant design related problems have surfaced during these tests; however, a braze deficiency in the manufacturing process of the 77.5:1 area ratio nozzle at the aft manifold resulted in three test aborts due to excessive hydrogen leakage caused by tube failures at this manifold location. Consequences of a failure of this type during flight conditions could result in an engine shutdown. Criteria and procedures have been developed for modification and repair of similar conditions on other existing nozzles, and during subsequent testing the failures have not repeated. Although the flight engine configuration system testing previously summarized has been trouble free of any significant design issues, several component failures have occurred during other testing conducted during this timeframe on similar units to require redesign, recertification, and retrofit to the flight engines at KSC. Failures of the nozzle hydrogen feed line, main fuel valve housing, and high pressure oxidizer turbopump turbine seal package have required modification to these parts and this work has now been completed. Requalification testing is still in progress.

Certification test results obtained todate have been quite successful with approximately two-thirds of the required testing completed. The 20,000 seconds of required operation has been split into four (4) test series involving 13 firings each at conditions enveloping the specification requirements. Certification has been limited to 100% of rated thrust conditions with one test in each series demonstrating 102% of rated thrust operation. A minimum of two abort mission thrust profiles are required with firing duration extending up to 823 seconds. These certification tests must be successfully completed on two (2) engines with each engine completing two (2) of the defined test series. Each test series must accumulate a minimum of 5,000 seconds of operation. Figure 6 shows engine serial number 0009 during certification testing. All tests conducted todate have demonstrated specific impulse above minimum requirements with an average of 454.5 seconds. This specific impulse is representative of performance measured on the flight engines during acceptance testing.

The NASA has imposed very specific requirements on the certification testing with defined criteria for successful completion. A summary of these criteria is presented for reference: (1) If a non-catastrophic shutdown does occur that is the result of flight hardware, flight software, or engine flight performance, and would have resulted in a mission abort, then that 5,000-second certification series must be rerun in its entirety. Any catastrophic failure will require overall certification reappraisal. (2) Any failure between 5,000 seconds and 10,000 seconds which is clearly time or cycle dependent (fatigue) can be accepted without a penalty certification if the overall engine certified life is lowered appropriately, including factors of safety for scatter. (3) Inspection and replacement of components between tests will be permitted, provided the basis is documented as a lien against the certification, and post-flight procedures of the same kind are made mandatory requirements. (4) A premature shutdown prior to SRB ignifiton will not require engine recertification. (5) Testing will be accomplished with flight redline policy. (6) Failures of instrumentation that would be detected by flight software reasonableness tests which would result in elimination of that redline for the remainder of that test are not considered a certification test failure.

Through development testing and the certification test series completed todate, approximately 65,000 seconds of single engine firing exposure has been accumulated as illustrated in Figure 7. By the end of this year, it is estimated that this test exposure will be extended to 80,000 seconds. Figure 8 breaks down the engine operating time for each thrust level range. Note that approximately 35,000 seconds have been accumulated at 100% of rated thrust or higher, with the majority of this time achieved during the last twelve months. During March of this year a limited amount of engine data was obtained at 109% of rated thrust conditions. The engine operating conditions at this thrust level were as expected and summarized in Figure 9. It is significant that this testing, although of limited duration, was achieved on engine serial number 2004 after successfully completing in excess of 10,000 seconds of operation during certification testing for the first launch. Further testing is planned on this engine at the higher thrust levels to obtain early insight into engine component operating parameters and to potentially provide abort capability for early Shuttle flights at increased thrust.

COMPARISON OF ENGINE DEVELOPMENT PROGRAMS

Although the requirements differ considerably between the SSME and engines used in the Saturn Program, specifically in areas of performance and reusability, it is instructive to relate the development activities where possible to assess progress and readiness for flight. In comparing prior engines with the SSME, the J-2 is probably the best choice since both use LOX/hydrogen as propellants, are in the 200 - 500K thrust class, and provide throttling capability.

The J-2 was a highly successful development effort of the Marshall Space Flight Center and the Rocketdyne Division of Rockwell International. The J-2 engine that powered the first manned Saturn Vehicle AS-205, on October 11, 1968, was rated at 200K thrust with mixture ratio control providing a thrust range from 225K to 175K. This engine, serial number 2033, was built, acceptance tested on 7/28/65, and certified three years before that flight. Certification consisted of a PFRT, FRT, and a Qual I test program. Development and certification testing of the 200K J-2 engine, as used for the first manned Saturn Vehicle, was terminated with completion of Qual I. The J-2 certifications (PFRT, FRT, and Qual I) were each conducted on a new engine, with accumulated certification operating time required ranging from 2,000 - 3,000 seconds. Two certification engines are used in the SSME program with each engine required to accumulate 10,000 seconds of operation within the criteria specified in the preceding section. Figure 10 summarizes and compares the results of the certification tests conducted on the J-2 and SSME engines with SSME data current through 3/26/80. The comparison includes the number of tests conducted, mission simulations, and success ratios (Px).

Each of these development programs were structured to focus on the type of testing critical to successful development within the resources available. With the technical complexity of the SSME, and a different budget environment than existed in the Saturn Program, the emphasis has been an efficient test planning and rigorous design analysis to minimize test requirements and program costs. Figure 11 depicts the engine operating time accumulated as a function of the tests conducted, one measure of efficiency and maturity growth, for the SSME, J-2 and F-1 engine programs. It appears from this data to be clear that the basic design, test planning, and overall conduct of the SSME development effort justifies fewer engine tests than Saturn experience. Figure 12 further compares the relative maturity of the SSME and J-2 engines as the first flight is approached. The mean time between premature shutdowns is the parameter plotted as a function of test history. For reference, the first Shuttle launch is assumed in November of 1980.

Cluster engine test progress is indicative of maturity growth and is compared on Figure 13 for the MPTA (SSME) and S-II Stage (J-2) programs. The history of the test programs is shown starting with the first test. The legend at the top of this figure will help differentiate between the J-2 and SSME experience. Note that for each cluster test conducted the actual firing duration achieved is portrayed relative to the duration planned. Several points can be made from this figure. (1) the first successful mission duration SSME cluster test was achieved on the ninth test, while the first successful mission duration J-2 cluster test occurred on the sixteenth test, and (2) consecutive successful mission duration tests on the J-2 engine cluster occurred only after three-fourths of the total program was complete. By comparison the SSME engine cluster achieved this milestone relatively early.

SPACE SHUTTLE MAIN ENGINE SUMMARY

Development and certification progress of the SSME appears to be progressing at a rate consistent with a planned Shuttle launch late this year. Testing conducted during the past year has generally yielded excellent results. The certification program has been expanded and is proceding on schedule with completion expected late this summer. At this time no significant design problems exist, with final verification testing on select components remaining. No significant system related problems have surfaced in the cluster testing conducted that would require engine modification. Progress of the SSME maturity growth, in general, parallels prior engine development experience and the outstanding flight success demonstrated on Saturn engines is expected with the Shuttle SSME's.

SPACE SHUTTLE MAIN ENGINE CHARACTERISTICS

● THRUST		
● SEA LEVEL	375K	(1,668,080 N)
● VACUUM	470K	(2,090,660 N)
● FULL POWER LEVEL	109%	109%
● CHAMBER PRESSURE	2995 PSIA	2065 N/cm^2
● NOZZLE AREA RATIO	77.5	77.5
● SPECIFIC IMPULSE (NOM)		
● SEA LEVEL	363.2	3562 $\frac{N\ sec}{kg}$
● VACUUM	455.2	4464 $\frac{N\ sec}{kg}$
● MIXTURE RATIO	6.0	6.0
● LENGTH	167"	424 cm
● DIAMETER		
● POWERHEAD	105" X 95"	267 X 240 cm
● NOZZLE EXIT	94"	239 cm
● LIFE	7.5 HRS	7.5 HRS
	55 STARTS:	55 STARTS

FIGURE 1

TEST STAND A-1-NSTL

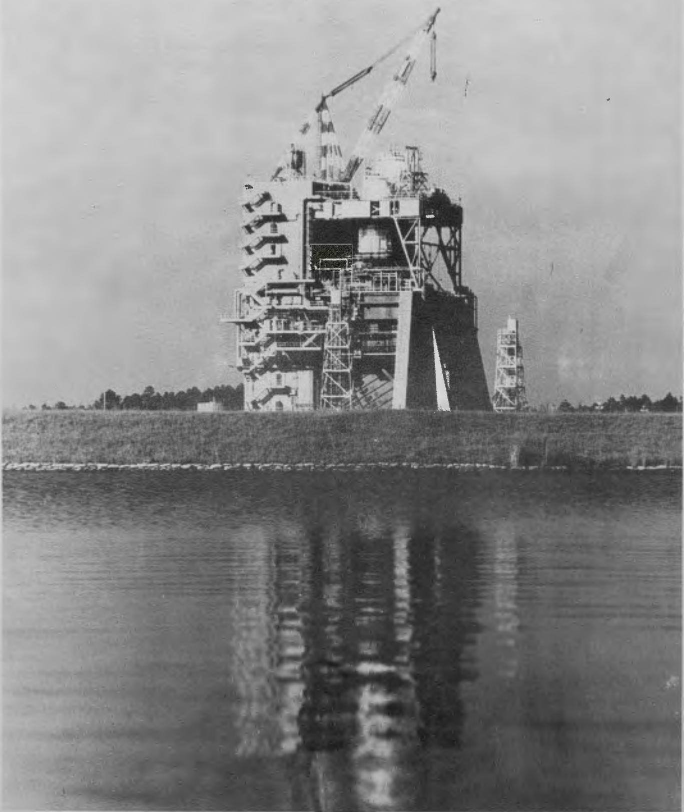
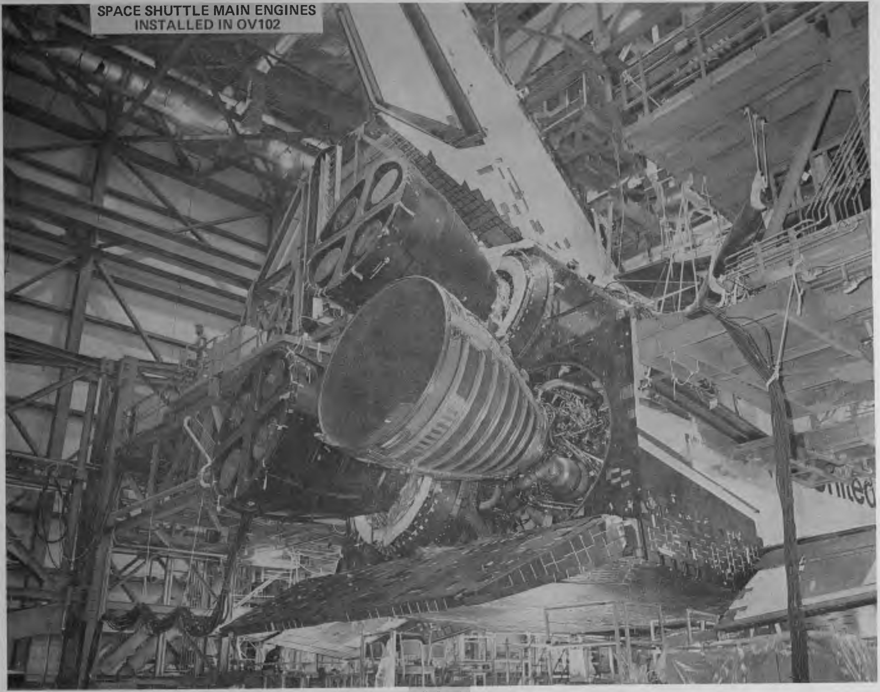


FIGURE 2



SPACE SHUTTLE MAIN ENGINES INSTALLED IN OV102

FIGURE 3

SSME MAJOR COMPONENTS

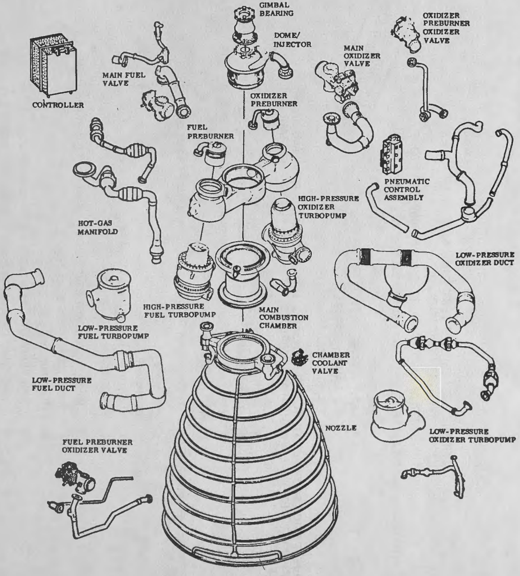


FIGURE 4

ENGINE RELIABILITY TEST SUMMARY
(FLIGHT CONDITIONS)
ENGINE SERIAL NOS. 2004, 2005, 2006, 2007, 0008 ,0009

FLIGHT ENGINE CONFIGURATION

- COUNTDOWN PROBLEMS AFTER PROPELLANT DROP RESULTING IN LAUNCH SCRUB.
  - 71 TESTS
  - 23.7 MISSIONS
  - 2 LAUNGH SCRUBS
  - .916 MISSION RELIABILITY

- ABORTS PRIOR TO SRB IGNITION. NEW PROBLEMS. (EXCLUDE REPEAT PROBLEMS ON SAME PART AND PROBLEMS WHICH ARE SCREENED OUT DURING FLIGHT ACCEPTANCE TESTING).
  - 71 TESTS
  - 23.7 MISSIONS
  - 3 ABORTS PRIOR TO SRB IGNITION
  - .873 MISSION RELIABILITY

  (IF TOTAL REDUNDANCY OF INSTRUMENTATION INVOLVED IN REDLINE PARAMETERS NOT REQUIRED, THEN –)
  - 71 TESTS
  - 23.7 MISSIONS
  - 1 ABORT PRIOR TO SRB IGNITION
  - .958 MISSION RELIABILITY

- LOSS OF REDUNDANCY DURING MISSION (CONTROL SYSTEM OR FLIGHT REDLINE PARAMETER).
  - 17711 SECONDS OF OPERATION
  - 11.35 MISSIONS
  - ONE "LEG" OF A REDUNDANT SYSTEM LOST 3 TIMES.
  - ONE "LEG" OF A REDUNDANT SYSTEM LOST EVERY 3.78 MISSIONS.

- ENGINE SHUTDOWN DURING MISSION. NEW PROBLEMS. EXCLUDE REPEAT PROBLEM ON SAME PART.
  - 17711 SECONDS OF OPERATION
  - 11.35 MISSIONS
  - 1 ENGINE SHUTDOWN
  - .912 MISSION RELIABILITY

- ENGINE MALFUNCTION SHUTDOWN POTENTIALLY AFFECTING VEHICLE SAFETY.
  - 17711 SECONDS OF OPERATION
  - 11.35 MISSIONS
  - 1 VEHICLE SAFETY ISSUE

FIGURE 5

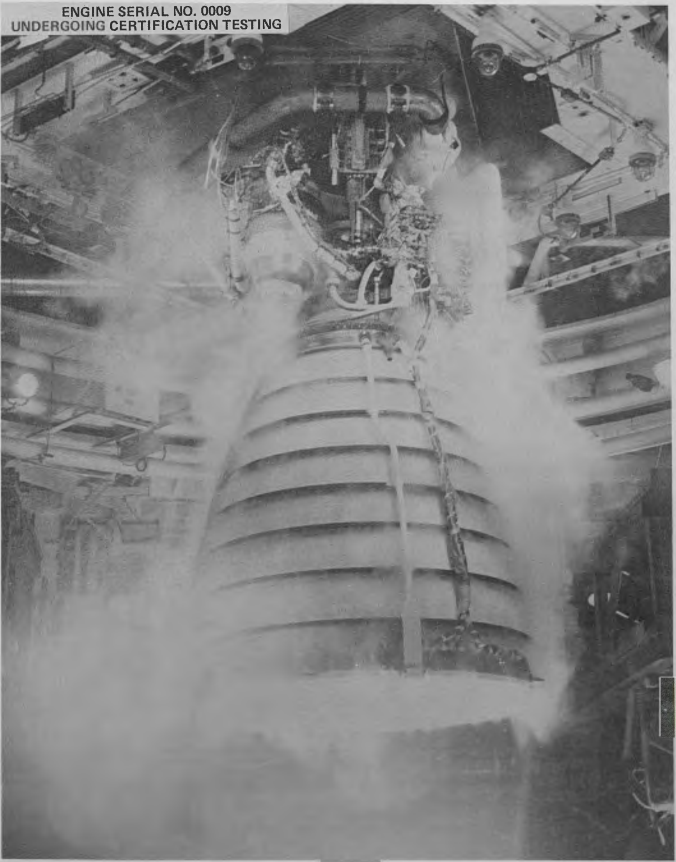


FIGURE 6

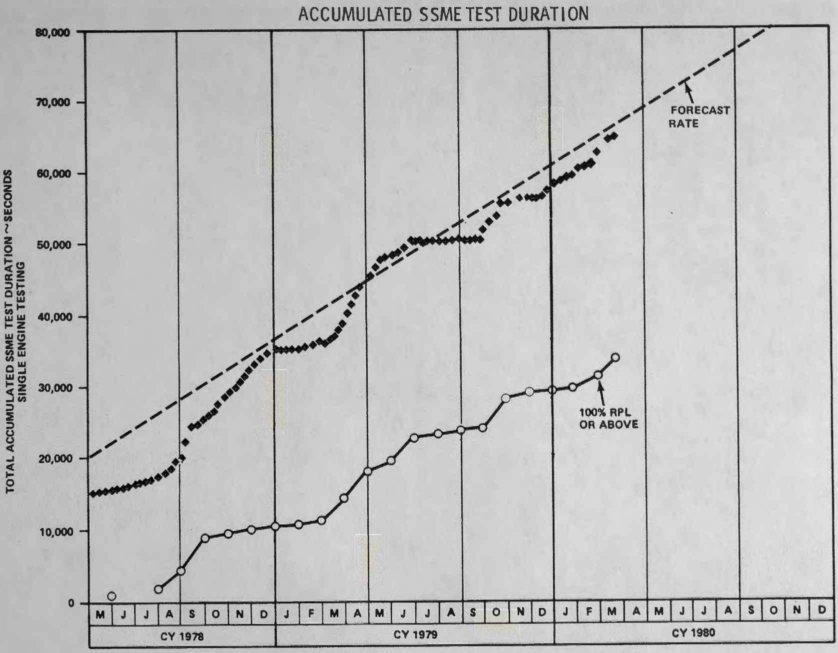

ACCUMULATED SSME TEST DURATION
TOTAL ACCUMULATED SSME TEST DURATION ~ SECONDS
SINGLE ENGINE TESTING
80,000
70,000
60,000
50,000
40,000
30,000
20,000
10,000
0
FORECAST
RATE
100% RPL
OR ABOVE
M J J A S O N D J F M A M J J A S O N D J F M A M J J A S O N D
CY 1978
CY 1979
CY 1980

FIGURE 7

SUMMARY OF ENGINE TEST POWER LEVEL OPERATION

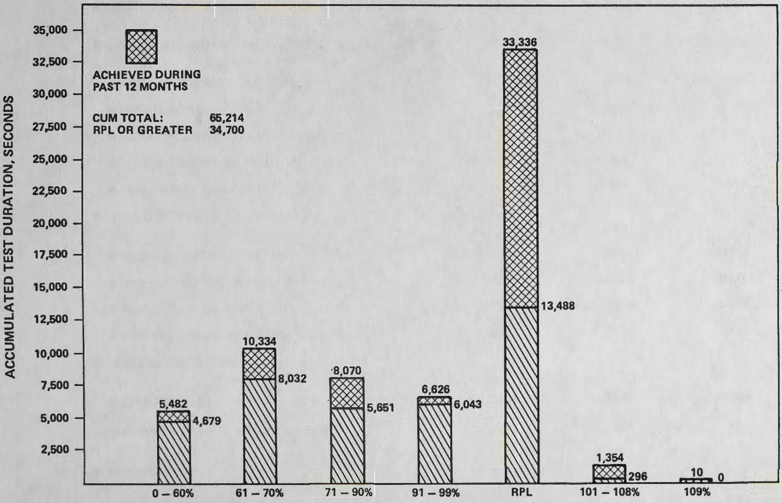


FIGURE 8

ENGINE 2004 PERFORMANCE SUMMARY

● PARAMETER			
● POWER LEVEL (% RATED THRUST)	100	106	109
● CHAMBER PRESSURE, PSI	2995	3166	3258
● TURBOPUMP SPEEDS			
● LOW PRESSURE OXIDIZER, RPM	4967	5225	5310
● LOW PRESSURE FUEL, RPM	15392	16290	16750
● HIGH PRESSURE OXIDIZER, RPM	28347	29440	30175
● HIGH PRESSURE FUEL, RPM	34924	36036	36745
● PUMP DISCHARGE PRESSURES			
● LOW PRESSURE OXIDIZER, PSIA	410	424	431
● LOW PRESSURE FUEL, PSIA	263	276	283
● HIGH PRESSURE OXIDIZER, PSIA	4289	4625	4795
● PREBURNER OXIDIZER, PSIA	7277	7935	8185
● HIGH PRESSURE FUEL, PSIA	6253	6641	6880
● OXIDIZER TURBINE DISCHARGE TEMPERATURE, ^{o}R	1260	1390	1385
● FUEL TURBINE DISCHARGE TEMPERATURE, ^{o}R	1742	1835	1860

FIGURE 9

CERTIFICATION SUMMARIES
PRIOR TO FIRST FLIGHT

	J–2 PFRT	J–2 FRT	J–2 QUAL 1	J–2 TOTAL	SSME PFC #1	SSME PFC #2	SSME* PFC #3A	SSME PFC #3B (THRU 3/26/80)	SSME TOTAL (THRU 3/26/80)
● TESTS									
● TOTAL CONDUCTED	12	18	23	53	13	13	2	4	32
● COMPLETED PLANNED DURATION	8	15	22	45	10	11	2	4	27
● PX (EXCLUDE FACILITY)	.727	.833	.956	.865	.769	.917	1.000	1.000	.871
● PX (EXCLUDE FACILITY, REPEAT FAILURE)	.80	.833	.957	.882	.909	.917	1.000	1.000	.931
● MISSION DURATION SIMULATIONS									
● TOTAL ATTEMPTED	4	4	5	13	11	10	2	4	27
● COMPLETED PLANNED DURATION	2	2	4	8	8	8	2	4	22
● PX (EXCLUDE FACILITY)	.50	.50	.80	.615	.727	.889	1.000	1.000	.846
● PX (EXCLUDE FACILITY, REPEAT FAILURE)	.50	.50	.80	.615	.889	.889	1.000	1.000	.917
● SHUTDOWNS THAT COULD INVOLVE VEHICLE SAFETY	1	2	1	4	0	0	0	0	0

*(ENGINE 0008. SINCE REALLOCATED TO CLUSTER ENGINE TESTING)

FIGURE 10

SUMMARY OF SSME, J-2 AND F-1 ENGINE ACCUMULATED TEST DURATION vs. TESTS CONDUCTED

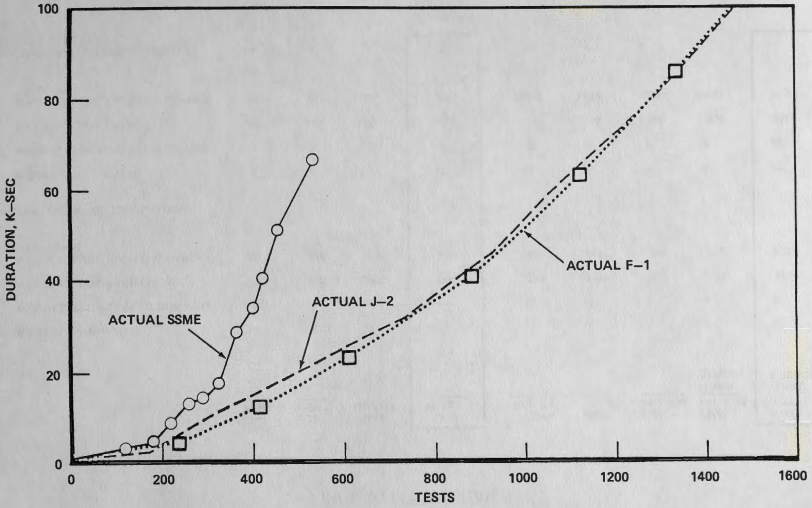
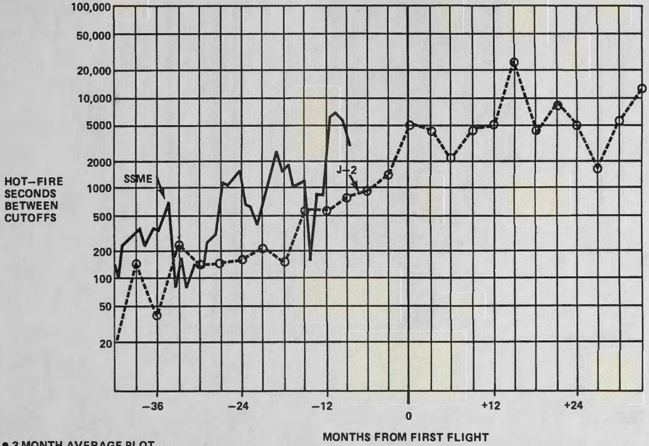


FIGURE 11

SSME AND J-2 ENGINE
MEAN TIME BETWEEN PREMATURE CUTOFFS



- 3 MONTH AVERAGE PLOT
- SINGLE ENGINE COMPARISON
- ENGINE RELATED PREMATURE CUTOFFS (ALL SINGLE ENGINE TESTS, VEHICLE TESTS, AND FLIGHTS; EXCLUDES CUTOFFS DUE TO FACILITY PROBLEMS)
- PLANNED FMOF NOV 1980

FIGURE 12

CLUSTER ENGINE PROGRAM COMPARISON
SSME AND J-2 ENGINES
MPTA AND S-II STAGE TESTING FOR INITIAL VEHICLE FLIGHT

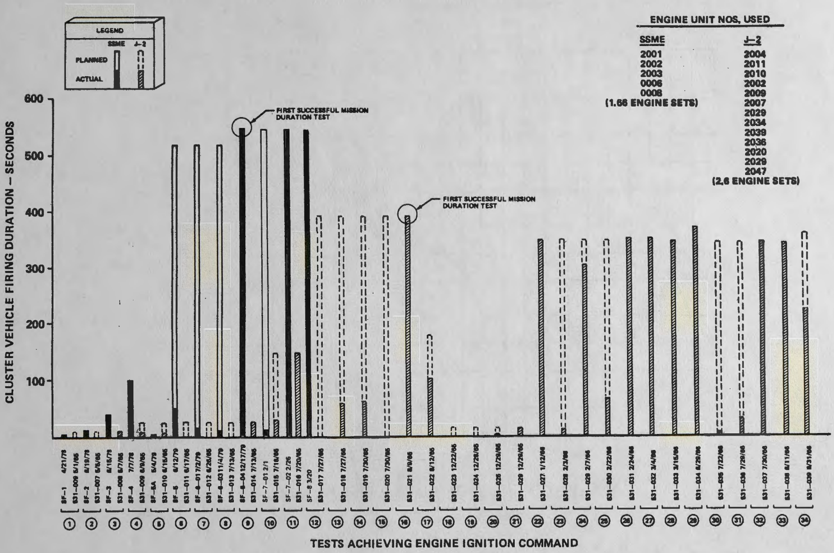


FIGURE 13